JUPITER ENTRY PROBE
FEASIBILITY STUDY FROM THE ESTEC CDF TEAM
HEAT FLUX EVALUATION & TPS DEFINITION

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ABSTRACT

In the context of the Technology Reference Studies of ESA's Science Payload and Advanced Concepts Office, a feasibility study on a "Jupiter Entry Probe" has been performed in October/November 2005 using the ESA Concurrent Design Facility with the aim to assess the feasibility of a 'minimum' Jupiter atmospheric entry probe.

This study is the first step in the assessment of an atmospheric Jupiter mission which aims to improve the knowledge on the Jupiter atmosphere and forms part of a larger effort to define a technology reference for Jupiter exploration.

The objectives of the JEP study were to:

- Perform an assessment of a technology reference atmospheric Jupiter mission
- Design the mission and the probe
- Produce industrial and operating costing
- Assess the programmatic and technology constraints to the mission

The paper provides a short introduction into the preliminary system definition followed by a discussion on the constraints on the aerothermodynamics and the required thermal protection system.

1. BACKGROUND

The objectives of the study were to assess the feasibility of a 'minimum' Jupiter atmospheric entry probe and to assess the impacts of such a probe on a potentially combined atmospheric and magnetospheric mission to Jupiter.

Atmospheric studies can be performed in two ways: in-situ analysis (e.g. entry probes) and remote sensing missions. The remote sensing instruments will require an orbit that is close enough to obtain sufficiently high spatial resolution. The conflict that can arise in the case of Jupiter is that if the orbit is lowered inside the main radiation belts the mission lifetime will be severely reduced, and the instrumentation performance could be significantly degraded.

The importance of having in-situ measurements is further given through the need to understand the atmosphere in greater depths. The composition up to 20 bar is believed to be reasonably understood. However, also here, in-situ data is too limited to support this assumption. Further, no quantitative results on He and H_2 in the deep well-mixed atmosphere are available and are only achievable by in-situ measurements.

A highly interesting scientific contribution would therefore be provided by an entry probe, penetrating the denser layers of the atmosphere, e.g. to a pressure of 100 bar. The Galileo probe showed that the entry point certainly isn't trivial: the probe entered in a dry region of the Jovian atmosphere, preventing the assessment of critical key measurements such as O/H ratios. This has shown the importance of a different entry strategy, e.g. in a higher latitude, and preferably by multiple probes.

To take full opportunity of a mission to Jupiter, in this study the atmospheric probe was assumed to be be accompanied by at least one Orbiter for magnetospheric measurements of the Jupiter magnetopause and magnetotail and/or for the study of the magnetic field in the polar regions.

2. MISSION DESIGN AND REQUIREMENTS

Two launch dates with relatively low delta-V demands were analysed: 2016 and 2023. The 2016 launch date was selected as baseline since it is the worst case in terms of delta-V and schedule between the two options. The transfer to Jupiter will incorporate swing-bys at Venus and Earth and takes almost six years.

The launch has been assumed to take place using the Soyuz 2-1b-Fregat from Kourou with direct insertion into interplanetary trajectory. This allows a spacecraft mass including adapter of 1350 kg.

Due to Jupiter's massive gravity field, the spacecraft will accelerate considerably as it approaches perijove. The consequence for the entry probe is that the inertial velocity at entry will amount to around 60 km/s with only a weak dependency on the hyperbolic entry velocity. As the Jupiter rotation period is less than 10 hours, the equatorial atmospheric rotation speed is almost 12.6 km/s. Therefore, the actual atmospheric entry velocity depends strongly on the entry location. For a prograde, near-equatorial entry, the relative entry velocity is thus reduced to about 47 km/s, for retrograde entry, it would be over 72 km/s and for a polar arrival 60 km/s.

The entry probe will be released from the orbiter 90 days before entering into Jupiter orbit. Thereafter the orbiter manoeuvres itself onto a safe, non-entry trajectory.

The baseline mission considered is based on one orbiter and one entry probe with a near-equatorial entry. The probe mass including system margin is about 300kg and is comparable to the Galileo probe (around 339 kg), which went only down to 20 bar. Two deep entry probes are currently found too mass constrained.

The following **system design requirements** were defined:

- Mission design shall include an Orbiter/Carrier to carry at least one atmospheric probe to Jupiter and release it.
- The mission design shall allow the probe to perform entry and descent into the Jovian atmosphere at near equatorial latitude (with an option of non-equatorial descent up to -30 deg / +30 deg latitude, if possible).
- In-situ atmospheric properties shall be measured down to an altitude corresponding to 100 bar and using a given Strawman payload.
- Data shall be transmitted in real time to the accompanying Orbiter.
- A multi-probe mission shall be achieved if mass allows.
- Preferred launch dates: 2016 (baseline) or 2023
- Launch vehicle: Soyuz 2-1b from Kourou

For the entry probe itself the following **design drivers** have been identified:

- Jupiter atmosphere
- High velocity entry
- Synchronisation of the Probe with the Orbiter
- Mass of the Probe.

Considerable uncertainties exist on the physical and chemical parameters, which call for high design margins.

The entry velocity cannot be reduced below a minimum value of ~ 47 km/s. This represents the major design issue of the mission as at such speed:

- Aerothermodynamic phenomena, in this regime and for the Jupiter atmosphere, are not well understood leading to large uncertainties in calculation of the heat fluxes/loads (RD[2])
- Very high aerothermodynamic heat fluxes are at the limit of present TPS technology capabilities
- High TPS mass fraction (~ 50%)

The interplanetary trajectory and the Orbiter final orbit fix the entry conditions of the Probe. This data, associated with the probe mass and shape and combined with the Jovian atmosphere model, provides the probe trajectory into the atmosphere and as a consequence, the heat fluxes and load, which size the TPS thickness and therefore the mass. Due to the high aerothermodynamic fluxes and load, the TPS mass is in the range of 50% of the total mass of the probe.

3. JUPITER ATMOSPHERE

The composition of the atmosphere of Jupiter has been measured by the Galileo probe between pressure levels of 0.51 bars and 21.1 bars (RD[1]). This composition is as follows (volume mixing ratio):

- H_2 0.86
- He 0.136
- CH_4 0.0018
- N_2 0.0007

The above composition is assumed to be rather uniform and it is valid for the troposphere and most of the stratosphere. The corresponding molar mass is 2.31×10^{-3} kg/mol.

Measurements performed during the swing-by of the Cassini spacecraft (RD[1]) reveal that the structure of the Jupiter atmosphere is relatively simple: a troposphere in convective equilibrium with a constant adiabatic lapse rate, topped by a well-defined tropopause at a minimum temperature of about 100 K, above which the temperature increases to a temperature of 160 K and remains practically constant in the region of maximum deceleration and heat fluxes during entry. Above that constant temperature region (roughly above 300 km) the temperature increases again.

Within this study a Jupiter reference atmosphere has been defined in two versions: a nominal Galileo-like atmosphere, and a cold atmosphere, with a:

- Physical interpolation of the Galileo ASI data
- Isothermal atmosphere at T = 110 K up to 100 mbar as observed by Cassini CIRS
- Constant temperature gradient between 100 mbar and 10 mbar
- Warm model: lower stratosphere with a constant temperature T=160 K above 10 mbar
- Cold model: lower stratosphere with a constant temperature T=150 K above 10 mbar

Note: coldest stratosphere, with smallest scale height is in principle the worst case for aerothermodynamic calculations.

4. HEAT FLUX CORRELATIONS

The Galileo mission has shown that high uncertainty exists in the calculation of heat fluxes for a Jupiter entry due to a combination of the very high entry velocity and insufficient chemical kinetic models of the H_2/He mixture in the interaction with the TPS. Figure 4-1 shows a comparison between the measured and calculated front shield recession for the Galileo probe.

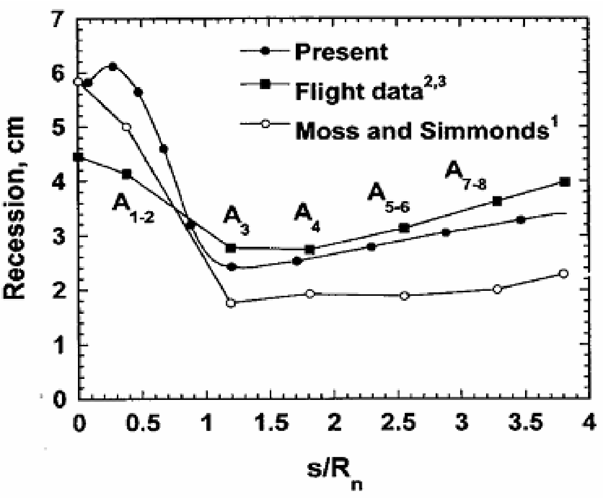

Figure 4-1: Comparison of recession profiles (RD[3])

RD[3] reports a recent attempt to explain the discrepancies. However, to date still not all phenomena can be considered explained. Therefore large uncertainty remains in the modelisation of the Jupiter entry by CFD.

Because of this intrinsic unreliability and of the short time available within the study, heat fluxes have been derived by building up correlation laws from CFD data provided through a literature survey. The correlations were built up using CFD calculation results from the Galileo development phase, and were validated through available data sets of Galileo in-flight measurements.

The correlations have been built for the convective and radiative heat fluxes without considering any radiation blockage from the ablation products in the shock layer.

For the **convective heat flux correlation** the following equation has been derived:

$$Q_c = 2004.2526 * \frac{1}{\sqrt{\frac{2*Rn}{0.6091}}} * \left(\frac{\rho}{1.22522}\right)^{0.4334341} * \left(\frac{V}{3048}\right)^{2.9978867}$$

- Q_c is the convective heat flux [kW/m^2],
- R_n is the nose radius [meter],
- ρ is the density [kg/m^3],
- V is the velocity [m/s].

For the **radiative heat flux correlations**, a selection of data sets has been taken to consider the impact of the atmospheric composition on the radiative emission of the flow-field. For a better agreement between data and correlation, data sets have been selected with a constant ballistic coefficient. The atmosphere has been considered to be composed of 89% H_2 and 11% He. The achieved correlation equation is as follows:

$$Q_r = 9.7632379^{-40} * (2 * R_n)^{-0.17905} * (\rho)^{1.763827469} * (V)^{10.993852}$$

where Q_r is the radiative heat flux [kW/m^2].

In order to **validate the correlations** above, comparison with Galileo in-flight measurements has been performed. The heat flux computed during the development phase has over-predicted the recession at the stagnation by 24% and under-predicted at the edge of the probe by 72% (Figure 4-1).

With the Galileo in-flight data (density and velocity) and considering the linearity of the recession versus heat flux, the Galileo heat flux has be evaluated and compared to the one computed with the correlations for a constant nose radius. Good agreement has been found as shown in Figure 4-2.

Radiative heat flux blockage of the shock layer radiation emission occurs due to the ablation products "injected" into the flow-field near the boundary layer. These are mostly C_2 and C_3 molecules that significantly absorb the radiation emission coming from the shock layer in front of the surface. The maximum blockage of the radiative heat fluxes according to RD[4] is around 60% at peak heating, whereas according to RD[5] it is reduced to about 44% due to spallation effects.

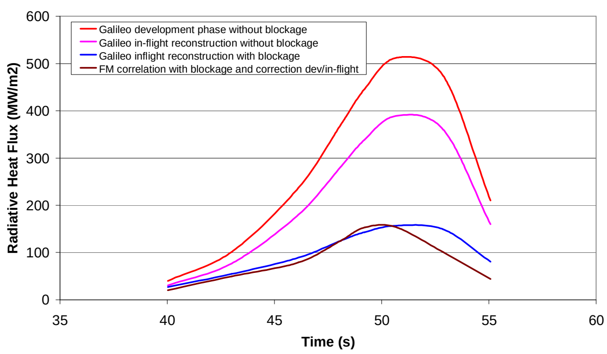

Figure 4-2: Comparison of radiative heat fluxes (Galileo in-flight, Galileo development and correlation)

Figure 4-2 shows the radiative heat fluxes without blockage calculated during the Galileo development (red curve) and reconstruction phases (pink curve). The blue curve represents the in-flight Galileo heat flux with the blockage due to the ablation phenomena.

Assuming the same flow conditions the correlation for the radiative heat flux has been recomputed and corrected in order to take into account the blockage factor and the over-estimation of the recession during the development phase. The updated correlation is also plotted in Figure 4-2 (brown curve).

5. PROBE DESIGN TRADE-OFF

A shape trade-off has been performed by RD[6] for three geometries with a half cone angle of 45˚, 50˚ and 55˚. The results have been compared in terms of:

- Heat flux at the stagnation point,
- Heat load at stagnation point and integrated over the surface,
- Heating rate during the entry along the front-shield surface.

The three geometries were compared for a mass of 250kg, a base radius of 88.9cm and a nose radius of 44.45cm. The entry point was considered to be at 450km with an initial velocity of 60km/s and a Flight Path Angle of -7.5˚.

The drag coefficients were considered constant over the entry phase and equal to 1.06, 1.25 and 1.38 for the 45˚, 50˚ and 55˚, respectively.

Configuration	Radiative Heat flux at stagnation point [MW/m^2]	Heat Load at stagnation point [MJ/m^2]	Integrated Heat Load over the body surface [MJ]
45˚ θ_c	619	4861	1028
50˚ θ_c	495	3785	1181
55˚ θ_c	426	3350	1439

Figure 5-1: Heatshield geometry trade-off

Figure 5-1 presents the results of the geometry trade-off. The 45˚ geometry has the maximum heat flux and heat load at the stagnation point. Nevertheless, it has the lowest integrated heat load over the entire front shield. This is due to different heating rates near the edge. The radiative heating on the edge is higher for the 55˚ geometry than for the 45˚ one. For 45˚ shape, the radiative heat flux is decreasing from the stagnation point towards the edge. Whereas for the 55˚ shape the heat flux reaches a minimum at S/Rb=0.3, but then increases again towards the rim to up to 80% of the stagnation point flux at some time during the entry phase.

In order to minimise the radiative heat flux and integrated heat load over the surface, the baseline shape for this study has been taken as a 45˚ half cone angle (as the case for Galileo). The base diameter and nose diameter are scaled functions of the probe mass.

6. AEROTHERMAL ANALYSIS

Two reference trajectories have been defined out of a large number of analysed cases: one equatorial and one non-equatorial entry. Different assumptions concerning the entry mass have been used for the trajectory computations.

A **non-equatorial entry** with a latitude of 15 degree has been studied. However, the calculated maximum heat flux values are above 900MW/m^2 (considering blockage effects). Since these loads are beyond the qualification level of the available ablator materials, a preliminary sizing approach would require a considerable extrapolation. Such an approach has been judged to be unreliable and not meaningful as no evidence is available that the considered material can sustain such load. This case has therefore not further been considered within the study.

The baseline assumptions used for an **equatorial entry** are the following:

- Entry Altitude: 450 km
- Entry Velocity: 47.4 km/s
- Entry Angle: -7.5˚
- Atmospheric model: Cold atmosphere (see above).

Two options have been studied depending on the final altitude and pressure which influences the mass budget of the probe and the decent module. The considered final pressures are respectively 100 bar and 40 bar with corresponding entry probe masses of 310 kg and 280 kg.

The radiative heat fluxes at the stagnation point in both options are presented in Figure 6-1 without and with blockage factor. The difference in mass (310kg/280kg) provides an increase of the radiative heat flux of 110 MW/m^2 (without blockage) and 40 MW/m^2 (with blockage).

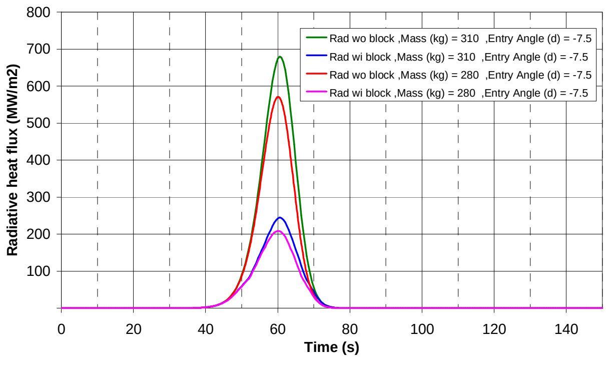

Figure 6-1: Heat Fluxes vs Time for an equatorial entry

For this study the 100 bar option was selected as baseline. Figure 6-2 shows the heat fluxes at stagnation point, mid-cone, edge and base point.

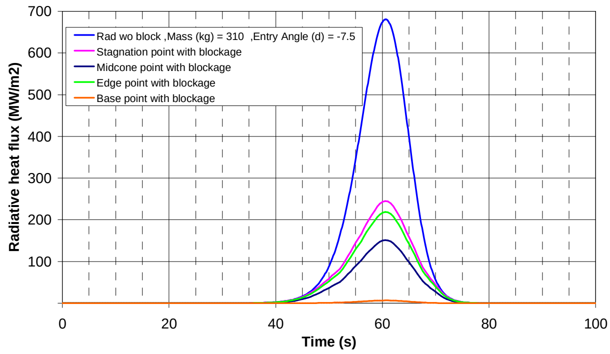

Figure 6-2: Heat Fluxes versus Time over the JEP surface for a Final pressure of 100 bars

7. BASELINE THERMAL PROTECTION DESIGN

The heat flux timelines assumed for the dimensioning of the TPS were derived from the above presented correlations and are presented in Figure 7-1.

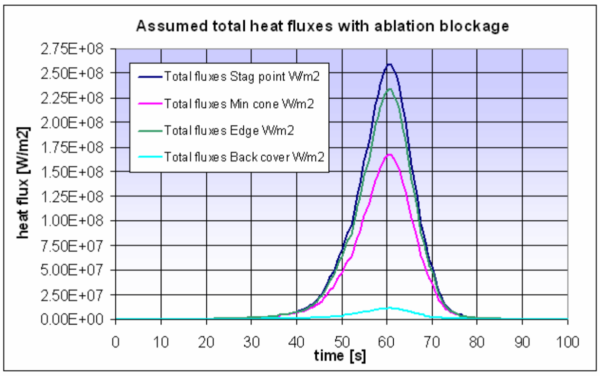

Figure 7-1: Assumed Baseline Heat Flux Timelines

On the radiative fluxes, blockage effects by the ablation are already considered by the ATD analysis. For the convective fluxes, references from Galileo indicate that they are almost entirely blocked by the ablation gases. However, as a conservative approach it has been assumed within this study that 10% of the convective heat fluxes are absorbed by the heatshield.

The base convective heating is assumed to be 1.25% of the stagnation point convective heating without blockage assumption. Similarly the base radiative heating is assumed to be 1.0% of the stagnation point radiative heating without blockage assumption. These assumptions take into account a 100% uncertainty.

These extremely high heat flux levels call for a very robust heatshield concept to secure the successful entry of the probe. Within the study a number of different concepts were investigated, out of which the most promising are described below.

1. Heat shield based on Carbon-Phenolic

- Having been applied and successfully demonstrated on the Galileo probe, carbon-phenolic is the only solution with a relevant heritage.
- Further experience with extreme heat fluxes comes from the application on the Pioneer-Venus Probes.

2. Heat shield based on Carbon-Carbon

- Generally, carbon-carbon materials demonstrate, compared to carbon-phenolic, a better ablative behaviour. However, a major drawback comes from its considerably higher thermal conductivity.
- Although extensive experience with carbon-carbon ablators exists from propulsion applications (nozzles), there is no direct relevant heritage for planetary entry mission.

3. Heatshield based on Carbon-SiC

- Similar remarks than for carbon-carbon apply.

4. Surface Protected Ablator (SPA)

- SPA is based on an ablative heatshield protected by an external rigid hot structure (e.g. C/SiC with venting holes) thereby achieving increased resistance against high dynamic pressure loads together with improved outer moldline stability.
- Flight experience was gained on the German MIRKA capsule. Maximum heat fluxes were about 1.3MW/m2. The concept is therefore immature in view of the expected entry environment for a JEP mission.

5. Porous ceramic heatshield filled with ablator

- This concept is based on ideas from DLR and NCSR Demokritos. It is based on a porous hot structure filled with ablative material. The ablator will thus cool the structure while the structure will provide a rigid structural resistance against the dynamic pressure loads.
- Successful tests have been performed for thruster applications. However, no entry experience has been achieved yet.

6. Sepcore concept

- The Sepcore concept, developed by SNECMA under ESA contract, uses a hot structure made of CMC. The ablator is mounted on top of the hot structure. The hot structure itself is insulated against the inner compartment by a lightweight insulator accommodated behind the hot structure.
- Main advantage is a considerable mass saving since the part of the ablator only used as insulator (without ablation) is replaced by a lightweight insulator. Allowable ablator backface temperatures are above 1000°C.
- Breadboard testing was successfully performed in air with maximum heat fluxes of $10 MW/m^2$.

In conclusion, a heatshield based on Carbon-Phenolic appears the most promising solution for a Jovian entry mission, having the most relevant heritage. In view of its high potential for mass reduction, the SEPCORE concept applying Carbon-Phenolic as ablator appeared to be the most promising solution.

As alternative options, a heatshield based on either Carbon-Carbon or Carbon-SiC ablators could be investigated. The shortcoming of these two ablators concerning their high thermal conductivity might be less stringent if applied within a Sepcore concept since high ablator backface temperatures are allowable.

A trade-off has been performed comparing a 'classical' heatshield (cold back structure) against a heatshield based on the SEPCORE concept. In both cases a carbon-phenolic ablator with a density of 1400kg/m^3 has been assumed. The results indicate that a TPS mass reduction of about 30% can be achieved with the SEPCORE concept. On the other side, the SEPCORE concept will lead to increased complexity due to the presence of the hot structure.

In consequence a Sepcore concept applying a Carbon-Phenolic ablator mounted on a hot structure made in C/SiC has been considered as baseline within this study.

The **preliminary sizing of the heatshield** has been performed using the ablation code ABLAT, which is integrated within the ESATAN thermal software. The material data of the considered ablator materials has been extracted from RD[7]. Due to the unavailability of a complete consistent characterisation data set, data sets from different test campaigns had to be combined.

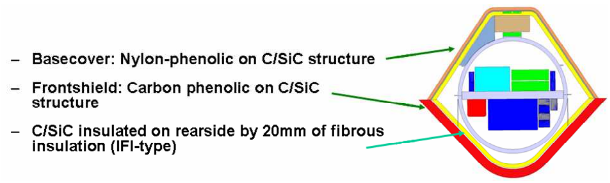

Figure 7-2: TPS Design Schematic

The principle of the TPS design is shown in Figure 7-2. The preliminary sizing of the TPS has been performed based on the following assumptions:

- Frontshield with fully dense carbon-phenolic ablator (density 1400 kg/m^3)
- Back cover with Nylon-phenolic ablator (density 1150 kg/m^3)
- Ablator attachment by ceramic screws
- Hot structure is insulated from the descent module by 20mm of fibrous insulation (IFI-type)
- Assumed temperature limit for the hot structure (ablator backface) is 1100°C
- Separation of the heatshield at 170sec.

The probe baseline design is composed of the descent module and the entry & descent system. The frontshield consists of a multi-layered TPS (Sepcore with carbon phenolic ablator RD[7]). The backcover also consists of a multi-layered TPS. The backcover supports one L-band antenna for communications during the entry up to the time when the back cover is jettisoned.

The preliminary analysis resulted in a required ablator thickness of 40mm for the stagnation point and 38mm for the lowest loaded point on the frustum. For mass calculation therefore, an average thickness of 39mm has been assumed over the front shield. A margin of 20% has been added on top, resulting in an ablator thickness of 47mm for the frontshield and 17mm for the backcover.

In addition, a 20% maturity margin was applied on the resulting mass. The overall margin policy applied for the TPS is shown in Figure 7-3. In view of the large uncertainties in the calculation of the heat fluxes, an even more generous margin might be found adequate.

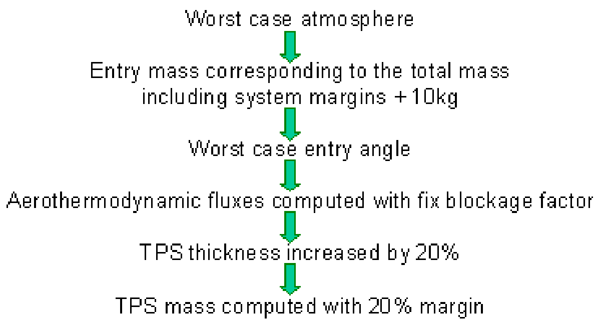

Figure 7-3: Margin policy for the TPS sizing

The **interface between the heatshield and the descent module** is sketched in Figure 7-4. The descent module is mounted on an interface ring made of C/SiC representing an integral part of the heatshield hot structure. This interface ring is supported by three longerons.

A simplified thermal analysis showed that the temperature increase at the skin of the descent module during the entry phase is about 40K, while the temperature increase within the descent module compartment is limited to about 3K.

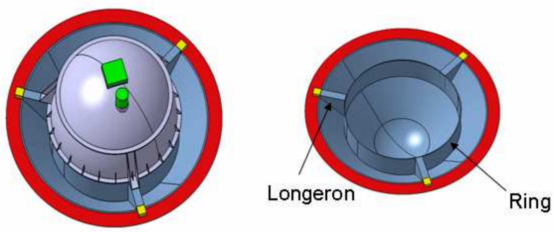

Figure 7-4: Interface b/w heatshield and descent module

The frontshield separation mechanism needs to be accommodated close to the descent module, away from the hot structure, in order to keep it below its temperature limit of about 50°C.

An additional MLI is required between the descent module and the heat shield in order to avoid cooling down of the probe after its separation from the mothership. This phase will last about 90 days. The MLI is considered to be mounted on the inside of the heatshield directly behind the fibrous insulation blanket required for the entry phase.

Thermal analysis shows that in addition some limited active heating might be required depending on the final detailed design. A heater power of several Watt provided by RHU is assumed.

After separation of the heatshield, the **thermal design of the descent module** has to guarantee the operational temperature limits of the module compartment through the hot Jovian atmosphere. This represents the design case for the insulation within the descent module.

In the descent thermal analysis it has been assumed that the heat exchange between decent module and the environment is predominantly by convection. The internal dissipation was assumed to be 300W.

A nano-porous Aerogel insulation blanket with a thickness of 20mm has been defined to be applied on the inner side of the decent module, due to their outstanding insulation performance when under atmosphere. A main driver of the temperature inside the descent module is its internal dissipation. Therefore, a further increase of the insulation thickness would not have any significant effect on the temperature inside the descent module.

In order to allow post-flight assessment of the heatshield performance a set of **ablation sensors** is assumed to be placed in the heatshield. The technology proposed is based on the sensors used for the Galileo probe consisting of analogue resistance ablation detectors (ARAD) and are assumed to be installed at different locations within the heatshield. The total number proposed is about 15.

8. VERIFICATION APPROACH

In defining the development plan for a Jupiter entry mission, the main issue is the qualification against the entry and descent loads, particularly, aero-thermodynamic and thermal protection.

The main design issues for a Jupiter entry probe are the peculiarity of the Jupiter environment (H_2-He atmosphere) and the high speed entry. As a considerable fraction of the probe mass goes into thermal protection, design of thermal protection needs to be as accurate as possible and acceptable design margins need to be implemented.

A reliable design of the TPS is based on:

- Accurate modelling of heat fluxes at entry,
- Reliable and well-known behaviour of the TPS material in the load range.

The Galileo experience has shown that the knowledge on phenomena incurring in the shock layer during a Jupiter entry is incomplete and models may be inaccurate by a significant factor.

Therefore validation of CFD models in a plasma facility capable to reproduce Jupiter entry conditions would be highly desirable. Unfortunately, representative test conditions appear not achievable in ground-based facilities and, rather paradoxically, the cost of building it would be comparable (or exceed) the cost of the whole mission, to the extent that it might be preferable to fly a probe into the Jupiter atmosphere as a validation of the CFD models. The cost of the facility for a single mission appears not justified.

In addition, later analyses of the experimental data collected during the Galileo flight, have shown shortcomings in the testing approach and in the facility capability to reproduce even a subset of the actual flight conditions.

For all these reasons alternative approaches have been investigated. The approach that best matches cost efficiency with validation looks to split the testing activities into two separate problems:

1. Partial validation of the aerothermodynamics models in a reduced flight envelope in existing facilities after dedicated modifications, and extrapolation of results to the high flight envelope.

2. Validation of TPS performance in radiative facilities (i.e. without interaction with flow) at expected (or even higher) flight heat flux levels.

Concerning aerodynamic facilities, a research among several existing European plasma tunnels has been performed. This overview has shown that:

- The Mach number range of these facility is far lower than the required one with a maximum in the order of 20.
- The maximum heat fluxes that can be generated on a TPS sample are about one order of magnitude lower than required (10-20 MW/m^2) with the significant exception of one high pressure facility that could reach heat fluxes up to 150 MW/m^2.
- The facilities cannot run with H_2-He as working plasma and modifications would be very significant including serious safety issues.

Other facility concepts as the Voitenko compressor were also looked at and may be potential alternatives.

It is not excluded that, for some facilities, modifications to allow hydrogen as working fluid are conceivable with an achievable flow range up to Mach 20. However, due to the short time available for the study, a more detailed assessment has not been possible.

Concerning TPS testing, the best approach found is to use focused high power lasers. Such facilities are used e.g. for testing of thermal barrier coatings on the internal walls of rocket engine combustion chambers where comparable heat flux levels are reached.

9. CONCLUSION

The study has shown that, for the given payload, a minimum Jupiter entry probe of about 300 kg can be designed reaching an altitude down into the atmosphere corresponding to a pressure of 100 bars.

The entry is from a hyperbolic approach trajectory. Release from the mothership takes place about 90 days before entry. Only near equatorial latitudes can be targeted as for higher latitudes the entry heat fluxes exceed the present capabilities of ablative thermal protection systems and the TPS mass fraction would reach values above 70%.

The probe design includes a generous margin for TPS design. This is required because a large uncertainty exists in the calculation of heat fluxes and performance of TPS in this thermal load range. Such uncertainties come from the fact that design and qualification will have to rely only on partial representation of the physical phenomena and on a somewhat reduced environment.

The TPS design and qualification is the most critical issue of the mission. The study did not find a completely satisfactory approach to derive aerothermal fluxes and to define the TPS testing, within the budget considered affordable for this mission. Therefore, margins will have to remain very high and the option of flying two identical probes may help reducing the risk.

Should the scientific community push for a Jovian Entry Probe mission, the technology development would have to be started early to select and develop a suitable TPS material based on the know-how on carbon-phenolic and carbon-carbon materials. In parallel, definition or modification of facilities for testing at heat fluxes in the 500 MW/m^2 range needs to be pursued.

Concerning aerothermodynamics, the construction of a facility dedicated to simulation of high speed Jovian entry is not conceivable. Therefore, efforts will have to concentrate on simpler facilities for partial verifications and deeper understanding of the Galileo flight data.

Finally, as a lesson learned from the Huygens mission, and following the Galileo approach, instrumentation for reconstruction of the flight heat fluxes and TPS behaviour shall be included as an engineering payload.

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